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OPERATIONAL CHARACTERISTICS OF RA-14 AVON TURBOJET ENGINE

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RESEARCH MEMORANDUM

OPERATIONAL CHARACTERISTICS OF RA-14 AVON TURBOJET ENGINE

By Joseph N. Sivo and William L. Jones

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SUMMARY

The windmilling and altitude starting characteristics for the RA-14 Avon turbojet engine were determined over a range of altitudes from 20,000 to 50,000 feet and flight Mach numbers from 0.4 to 1.0. Successful ignition and acceleration to idle speed were obtained up to an altitude of 33,000 feet at all Mach numbers investigated and at 45,000 feet for flight Mach numbers below 0.5 and above 1.0. The maximum operable altitude, where combustor blowout occurred, was approximately 80,000 feet.

Throttle bursts and wave-off-type accelerations were attempted at altitudes of 35,000 and 50,000 feet at a flight Mach number of 0.4. At 35,000 feet, successful accelerations to rated speed were obtained with both types of maneuvers. The wave-off-type accelerations at 35,000 feet, however, were characterized by a series of compressor surges during the accelerations. At 50,000 feet, all acceleration attempts terminated in surge.

Over-all performance of the engine with compressor-outlet bleed was determined at an altitude of 37,500 feet and a flight Mach number of 0.8. With 6-percent bleed flow at limiting engine temperature ratio, the net thrust decreased 14.2 percent, and the net-thrust specific fuel consumption increased 9.8 percent.

INTRODUCTION

An investigation of the altitude performance characteristics of the RA-14 Avon turbojet engine was conducted at the request of the Bureau of Aeronautics, Department of the Navy, in an altitude test chamber at the NACA Lewis laboratory. The over-all engine performance characteristics with a fixed-area exhaust nozzle are presented in reference 1, and the acceleration characteristics are presented in reference 2. The operational characteristics of the engine are presented herein. The operational characteristics investigated were

(1) Windmilling

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(2) Altitude starting and acceleration to idle

(3) Altitude operating limits

(4) Throttle bursts and wave-off-type accelerations with standard control

(5) Engine performance with compressor-outlet bleed

Windmilling and starting data were obtained over a range of altitudes 4074
from 20,000 to 50,000 feet and flight Mach numbers from 0.4 to 1.0. Throttle bursts and wave-off-type accelerations were investigated at altitudes of 35,000 and 50,000 feet and a flight Mach number of 0.4. Data on the effect of compressor-outlet bleed were taken at 37,500 feet and a flight Mach number of 0.8.

APPARATUS

Engine

The RA-14 Avon turbojet engine has a rated sea-level static thrust of 9500 pounds at an engine speed of 7850 rpm and a limiting turbine-outlet temperature of 1148° F (1608° R). The engine is equipped with a 15-stage axial-flow compressor, a cannular combustor with eight tubular liners, and a two-stage turbine.

At the suggestion of the manufacturer, the engine was operated to the rating of the RA-28 Avon turbojet engine, which has a rated sea-level static thrust of 10,000 pounds at an engine speed of 8000 rpm and a limiting turbine-outlet temperature of 1220° F (1680° R).

Limiting turbine-outlet temperature was determined by the manufacturer's four thermocouples. For this investigation, the engine was operated with a 2.36-square-foot (20.8-in. diam.) fixed-area conical exhaust nozzle.

The engine is equipped with variable compressor-inlet guide vanes; a two-position (open or closed), seventh-stage acceleration bleed port; and compressor-outlet bleeds for supplying aircraft service air. The inlet guide vanes are scheduled as a function of corrected engine speed. Below a corrected engine speed of 6100 rpm, the inlet guide vanes are closed. From 6100 to 7200 rpm, the guide vanes vary linearly with engine speed from full closed (25°) to full open (-10°). The acceleration bleed port is open below a corrected speed of 6200 rpm and closed above that speed. The compressor-outlet bleeds are operated in accordance with the needs of the aircraft installation.

Installation

The engine was installed in an altitude test chamber which consists of a tank 10 feet in diameter and 60 feet long divided into two compartments by a bulkhead. Air from the front compartment was ducted to the engine inlet through a bellmouth inlet and a venturi, which was used to measure the air flow. A labyrinth seal on the inlet duct was used to prevent leakage from the front to the rear compartment. The engine was mounted on a thrust-measuring platform in the rear compartment. The pressure and temperature in the front compartment and the pressure in the rear compartment were regulated to simulate the desired altitude flight conditions. A photograph of the engine installed in the test chamber is shown in figure 1.

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Instrumentation

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Instrumentation for measuring steady-state temperatures and pressures was installed at various stations throughout the engine. A schematic sketch of the engine showing station locations and pressure and temperature instrumentation is presented in figure 2. The transient instrumentation used in conjunction with the steady-state instrumentation is presented in table I.

Engine fuel flow during steady-state operation was measured with calibrated rotameters; engine speed, with remote-reading tachometers; and engine thrust, with a null-type thrust cell.

Multichannel oscillographs were used to obtain transient recordings of pressures, temperatures, fuel flow, throttle position, engine speed, and guide-vane and acceleration bleed positions during starting and throttle bursts (see table I). The transient equipment used had sufficiently rapid response to allow measurement of quantitative values.

PROCEDURE

For the windmilling and starting tests, the engine-inlet total temperature and pressure and the exhaust-nozzle static pressure were set to simulate a range of altitudes from 20,000 to 50,000 feet and flight Mach numbers from 0.4 to 1.0. A free-stream ram recovery of 100 percent was assumed. Prior to each ignition attempt, complete steady-state windmilling data were obtained. With the engine windmilling, the engine throttle was advanced to the idle position. A total of 45 seconds was allowed for ignition. If ignition did not occur in 45 seconds, the throttle was retarded and the engine was allowed to windmill 3 to 4 minutes before ignition was attempted again. Whenever ignition and complete propagation occurred, as determined by readings from a thermocouple in each combustor liner, the engine was allowed to accelerate to idle speed. If propagation of flame was not complete, the throttle was retarded in preparation for another ignition attempt.

Throttle burst and wave-off-type acceleration data were obtained at 35,000 and 50,000 feet at a flight Mach number of 0.4. A throttle burst consisted of an acceleration from idle to rated speed along the engine control schedule with the advance of the engine throttle from the idle to rated position completed in 1 second or less. The wave-off-type acceleration consisted of a deceleration from rated to idle speed followed immediately by a throttle burst with all throttle movements completed in 1 second or less. In general, the engine was allowed approximately 3 minutes to reach equilibrium temperature conditions prior to a throttle-burst acceleration or wave-off-type maneuver. 4074

Compressor-outlet bleed data were obtained at an altitude of 37,500 feet at a flight Mach number of 0.8. Engine corrected speeds of 6660, 7300, 7850, 8400, and 8580 rpm were set. At each corrected engine speed, a range of bleed flows was run from zero flow to either maximum bleed flow, as governed by the bleed-port size, or rated turbine-outlet temperature.

RESULTS AND DISCUSSION

Windmilling

Windmilling data (table II) were obtained over a range of altitudes from 20,000 to 50,000 feet and flight Mach numbers from 0.4 to 1.0 and are presented in figures 3 to 5. Corrected engine windmilling speed is presented as a function of flight Mach number in figure 3. A single curve defines the corrected speed for the range of conditions investigated. At a flight Mach number of 0.6, the engine windmilled at approximately 29 percent of rated speed (7850 rpm).

Curves of corrected engine air flow, compressor total-pressure ratio, and compressor total-temperature ratio against corrected engine windmilling speed are presented in figure 4. Over the range of corrected engine speeds associated with windmilling conditions, satisfactory generalization of all these parameters was obtained. At corrected speeds below 2800 rpm, a pressure loss occurred across the compressor. Corrected windmilling drag against corrected air speed is presented in figure 5. The data scatter here is a result of the difficulty in determining net thrust accurately at the low thrust levels involved.

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Ignition and Acceleration to Idle

The altitude starting limits for a range of flight Mach numbers are presented in figure 6. Each symbol on the figure denotes two or more ignition attempts. The maximum altitude at which ignition was attempted was 50,000 feet. For Mach numbers below 0.5, ignition was obtained up to this altitude. As the Mach number was increased to 0.7, the maximum altitude for ignition decreased to approximately 33,000 feet. A further increase in Mach number to 1.0 increased the maximum altitude for ignition to 45,000 feet. This trend in ignition limits has been observed on other engines. The lowering of the ignition limits as the flight Mach number was increased is attributed to the increase in combustor velocities that occurred with increasing Mach number in this region. (As noted in fig. 4, the air flow increased without a significant increase in compressor pressure ratio up to a Mach number of about 0.7 (corrected windmilling speed of 2700 rpm).) As the Mach number was further increased, the increase in pressure rise across the compressor offset the increase in velocity, resulting in a more favorable condition for ignition.

The following table lists the combustor-inlet conditions present at points along the ignition envelope:

Altitude, ft	Free-stream Mach number, M_0	Compressor-outlet total pressure, P_3	Compressor-outlet total temperature, T_3	Combustor velocity, V_b
50,000	0.5	271	420	44.7
35,000	.6	595	433	54.7
32,500	.7	755	460	67.8
33,000	.8	875	482	81.4
38,000	.9	895	505	97.8
45,000	1.0	820	546	120.0

A region of incomplete propagation existed at an altitude of 50,000 feet at a flight Mach number of approximately 0.45.

An oscillograph trace of an altitude start is presented in figure 7 to illustrate how parameters such as engine speed, engine fuel flow, turbine-outlet temperature, compressor-inlet and -outlet pressures, ambient exhaust pressure, and fuel manifold pressure vary during a typical altitude start and acceleration to idle speed.

Ignition, propagation, and acceleration times obtained from traces
similar to that of figure 7 are presented in figure 8 for various flight
conditions. At an altitude of 30,000 feet (fig. 8(a)), an increase in
flight Mach number from 0.4 to 0.8 caused the ignition time to increase
from 4 to 8 seconds, the propagation time (determined by observation of
thermocouples located at the combustor outlet) to decrease from 2.5 to
1.0 seconds, and the time required to accelerate the engine to idle
speed to decrease from 38.5 to 13.0 seconds. The time required to es-
tablish fuel manifold pressure for this range of Mach numbers was ap- 4074
proximately 4.0 seconds. Therefore, at low flight speeds, the ignition
time (time elapsed from when the throttle was advanced until ignition
occurred) was primarily a function of the time required to establish fuel
flow. As the Mach number was increased to 0.8, an additional 3 seconds
over the time required to establish fuel flow was required for ignition.

The effect of altitude on these times at a Mach number of 0.4 is presented in figure 8(b). Ignition time increased slightly with altitude. However, the major portion of the time required for ignition can be attributed to the time required to obtain fuel manifold pressure (establishment of fuel flow) since ignition occurred almost immediately after the fuel pressure reached its steady-state pressure value. The acceleration time increased from 25.5 to 62 seconds when the altitude was increased from 20,000 to 40,000 feet. As previously mentioned, propagation was incomplete at an altitude of 50,000 feet; acceleration was therefore not possible at this flight condition. It was determined that ignition, propagation, and acceleration were possible at 45,000 feet and a flight Mach number below 0.5 and above 1.0.

Altitude Operating Limits

The operational limits of the engine with the rated exhaust nozzle are presented in figure 9. The operational limits presented are the engine speed for limiting turbine-outlet temperature, the engine speed for "idle" throttle position, and the windmilling speed as functions of altitude. For altitudes above 20,000 feet, limiting turbine-outlet temperature occurred below rated engine speed (7850 rpm). Increasing the altitude from 20,000 to 80,000 feet necessitated a 11-percent decrease in engine speed in order to maintain limiting turbine-outlet temperature. The idle-speed line intersected the limiting-temperature line at approximately 81,000 feet. In this region engine blowout occurred, but it was not determined whether or not the engine blowout was preceded by compressor surge.

Throttle Bursts and Wave-Off-Type Acceleration

Throttle bursts, which consisted of accelerations from idle speed to rated speed, and wave-off-type accelerations, which consisted of decelerations from rated to idle speed followed by accelerations to rated speed, were obtained at two altitudes. All throttle movements were completed in 1 second or less. A typical multichannel trace of a throttle burst showing how the various engine parameters varied during the engine acceleration is presented in figure 10.

4074 The compressor-inlet-guide-vane position and the acceleration path taken by the engine in terms of compressor total-pressure ratio and corrected engine speed are presented in figure 11. Data are shown for two throttle bursts and one wave-off maneuver.

The guide-vane position during the accelerations differed from the average production guide-vane-position schedule (preferred schedule) for all acceleration attempts. The engine speed at which the bleed ports closed was also different for each acceleration attempt. The deviations from the preferred schedule may have been due to either improper temperature compensation in the control system or to differences in the temperature and pressure conditions surrounding the engine and control system from what was anticipated for flight conditions. A hysteresis also occurred in the guide-vane schedule, as indicated by the guide-vane schedule obtained during the wave-off-type maneuver (path ABCDEA).

Surge-free acceleration was obtained for a rapid acceleration from idle to rated speed when the engine was operated at idle speed for at least 3 minutes prior to an acceleration attempt. When the idling period was reduced, one cycle of surge occurred at a corrected engine speed of 7080 rpm as the engine accelerated to rated speed.

For the wave-off-type acceleration, engine surge occurred at corrected speeds from 5800 to 7500 rpm, while there was no surge from 7500 rpm to rated speed. The acceleration was characterized by a series of one or two cycles of surge followed by recovery as the engine accelerated.

The differences in the total-pressure ratio during surge for the throttle bursts and for the wave-off-type maneuver were, in part, due to differences in the inlet-guide-vane schedules mentioned previously.

The shift in the steady-state operating lines was also a result of differences in inlet-guide-vane schedules for the different acceleration attempts. The operating lines for their corresponding guide-vane schedules were determined from previously run unpublished data on the effect of inlet-guide-vane variation.

The time required for the engine to accelerate from idle to rated speed at 35,000 feet and a flight Mach number of 0.4 with no surge when stall did not occur was 8.0 seconds. This time increased to 8.4 seconds when one cycle of surge occurred and to 10.3 seconds when several cycles of surge occurred.

At an altitude of 50,000 feet and flight Mach number of 0.4, all attempts to accelerate the engine from idle speed terminated in surge from which the engine could not recover without throttle manipulation. 4074
At this altitude the engine did not accelerate during surge. All wave-off attempts also terminated in surge. From the transient traces of fuel flow obtained during acceleration at an altitude of 35,000 feet, it appeared that, when surge occurred and compressor-outlet pressure decreased, the engine fuel control reduced fuel flow to a value where surge recovery was possible. However, at 50,000 feet, when surge occurred, the fuel control did not reduce the fuel flow enough to allow com-surge recovery. A successful surge-free acceleration was made at 50,000 feet from 7400 rpm to rated speed. The time required for acceleration at this condition was 6.0 seconds as compared with 1.4 seconds required at 35,000 feet for the same condition.

The maximum exhaust-gas temperature reached was approximately 1650° R for all the acceleration attempts made during this investigation. A more complete analysis of throttle bursts and wave-off accelerations is presented in reference 2.

Compressor-Outlet Bleed

At each of five corrected engine speeds, a range of compressor-outlet bleed-air flows was run by varying the bleed-port area. The data obtained are tabulated in table III. The figures presented herein were obtained by cross-plotting the tabulated data.

The maximum bleed-flow ratio (ratio of bleed air flow to engine-inlet air flow) as limited by either the size of the bleed ports or limiting turbine-outlet temperature is presented in figure 12 as a function of corrected engine speed. The maximum bleed flow obtained was 7.1 percent at a corrected speed of approximately 8300 rpm.

The over-all engine performance characteristics for various bleed flows are plotted in figure 13 for an altitude of 37,500 feet and a flight Mach number of 0.8. With limiting engine temperature ratio ($T_9/T_2 = 3.63$), an increase in compressor-outlet bleed flow from zero to 6 percent necessitated a 2-percent reduction in engine speed (fig. 13(a)). At limiting temperature ratio, the fuel flow (fig. 13(b)) decreased 4.2 percent as a result of the decreased engine speed when the bleed flow was increased from zero to 6 percent.

A loss in net thrust of 14.2 percent occurred when the bleed flow was increased from zero to 6 percent at limiting temperature ratio (fig. 13(c)). This loss was a result of both the engine speed reduction required to maintain rated temperature and the loss in engine air flow overboard through the bleed ports. The reduction in engine speed reduced engine pressure ratio and air flow. The bleed flow lost overboard necessitated an increase in the extracted turbine work per pound of gas flowing through the turbine, that is, increased turbine pressure ratio which further reduces the engine pressure ratio.

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Because the net thrust and engine fuel flow decreased 14.2 and 4.2 percent, respectively, when the bleed flow was increased to 6 percent at limiting temperature ratio, the net-thrust specific fuel consumption increased approximately 9.8 percent at this condition (fig. 13(d)).

SUMMARY OF RESULTS

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From an investigation of the windmilling and starting characteristics of the RA-14 Avon turbojet engine, the compressor pressure ratio, temperature ratio, and air flow generalized with corrected engine speed for the range of flight conditions presented. Successful ignition and acceleration to idle speed were obtained at an altitude of 33,000 feet at all Mach numbers investigated and at an altitude of 45,000 feet for flight Mach numbers below 0.5 and above 1.0.

In order to maintain limiting turbine-outlet temperature with the rated fixed-area exhaust nozzle installed on the engine, a reduction of 11 percent in engine speed was required as the altitude was increased from 20,000 to 80,000 feet. The maximum operable altitude, where combustor blowout occurred, was approximately 80,000 feet.

With the standard engine control, successful accelerations from idle to rated speed were obtained at 35,000 feet at a flight Mach number of 0.4 for both throttle-burst and wave-off-type accelerations. The wave-off-type accelerations were characterized by a series of compressor surges during the accelerations. At 50,000 feet all acceleration attempts from idle speed terminated in compressor surge.

At an altitude of 37,500 feet and a flight Mach number of 0.8, a 6-percent compressor-outlet bleed flow at limiting engine temperature ratio resulted in a 14.2-percent decrease in net thrust and a 9.8-percent increase in net-thrust specific fuel consumption.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, April 13, 1956

APPENDIX - SYMBOLS

F_j	jet thrust, lb
F_n	net thrust, lb
M	flight Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft abs
p	static pressure, lb/sq ft abs
S_l	large-slot fuel manifold pressure, lb/sq in.
S_s	small-slot fuel manifold pressure, lb/sq in.
sfc	net-thrust specific fuel consumption, w_f/F_n, (lb/hr)/lb thrust
T	total temperature, ^{o}R
V	velocity, ft/sec
w	weight flow, lb/sec or lb/hr
δ_2	ratio of absolute total pressure to absolute static pressure of standard NACA atmosphere at sea level
θ_2	ratio of absolute total temperature to absolute static temperature of standard NACA atmosphere at sea level

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Subscripts:

a	air
B	bleed air
b	combustor
f	fuel
0	free stream
1	air-flow measuring station
2	compressor inlet
3	compressor outlet

4 turbine inlet

5 turbine outlet

9 exhaust-nozzle inlet

10 exhaust-nozzle outlet

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REFERENCES

1. Sivo, Joseph N., and Jones, William L.: Preliminary Altitude Performance Data for the RA-14 Avon Turbojet Engine. NACA RM E55K07a, 1955.

2. Russey, Robert E.: Altitude Acceleration Investigation of the RA-14 Avon Turbojet Engine. NACA RM E56C01, 1956.

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TABLE I. - INSTRUMENTATION

Parameter	Engine station	Steady-state instrumentation	Transient instrumentation
Engine throttle	--	Manual	Limit switches at idle and full-throttle positions
Engine fuel flow	--	Calibrated rotometer	a-c Output of flow meter rectified to d-c voltage proportional to fuel flow
Inlet total pressure	2	Average of 18 total-pressure probes	Aneroid-type pressure sensor with strain-gage element
Compressor-outlet total pressure	3	Average of 20 total-pressure probes	Aneroid-type pressure sensor with strain-gage element
Altitude static pressure	10	Average of 4 lip static-pressure probes at exhaust-nozzle exit	Aneroid-type pressure sensor with strain-gage element
Exhaust-nozzle-inlet total pressure	9	Average of 16 total-pressure probes at exhaust-nozzle inlet	Aneroid-type pressure sensor with strain-gage element
Exhaust-nozzle-inlet total temperature	9	Average of 22 thermocouples	Single thermocouple with electronic network to compensate for thermocouple lag
Inlet-guide-vane position	--	Potentiometer connected to master guide vane	Same as steady-state instrumentation
Interstage bleed position	--	Two total- and one static-pressure probes in bleed duct	Bourdon type pressure sensor completing electric circuit
Compressor-outlet bleed	--	Two ducts with two total- and one static-pressure probes in bleed duct	------------------------------
Engine speed	--	Engine tachometer generator and electronic pulse counter	Engine tachometer generator, a-c rectified to d-c voltage proportional to speed
Large slot, main fuel pressure	--	Bourdon gage	Twisted-tube pressure sensor with variable reluctance bridge
Small slot, pilot fuel pressure	--	Bourdon gage	Twisted-tube pressure sensor with variable reluctance bridge

TABLE II. - WINDMILLING PERFORMANCE DATA

Altitude, ft	Free-stream Mach number, M_0	Corrected engine speed, $N/\sqrt{\theta_2}$, rpm	Compressor-inlet total pressure, P_2, $\frac{lb}{sq\ ft\ abs}$	Altitude static pressure, p_0, $\frac{lb}{sq\ ft\ abs}$	Compressor-inlet total temperature, T_2, $^{\circ}R$	Compressor outlet		Corrected windmilling air flow, $\frac{w_{a,2}\sqrt{\theta_2}}{\delta_2}$, lb/sec
						Total pressure, P_3, $\frac{lb}{sq\ ft\ abs}$	Total temperature, T_3, $^{\circ}R$	
20,000	0.39	1468	1085	978	466	1046	467	18.29
	.60	2342	1242	978	481	1206	491	28.52
30,000	0.39	1433	704	632	422	680	422	17.38
	.59	2277	802	634	445	773	445	26.89
	.71	2802	880	631	454	876	474	33.46
	.78	3126	956	638	468	1004	498	38.10
35,000	0.58	2177	632	502	426	606	435	26.31
	.61	2301	639	497	421	604	421	28.78
	.71	2782	699	501	425	691	446	32.97
	.78	2975	751	504	445	772	469	34.59
	.80	3127	762	499	444	792	474	37.45
	1.09	4078	945	502	474	1280	547	51.99
	1.00	4201	945	506	475	1324	551	53.61
40,000	0.41	1409	440	391	411	419	419	17.82
	.41	1425	443	394	408	456	408	16.65
	.49	1855	469	398	415	451	430	22.09
	.59	2193	508	397	423	474	423	28.24
	.61	2247	508	394	424	479	424	26.17
	.72	2784	560	397	434	543	452	32.24
	1.00	4216	748	394	475	1057	553	53.66
45,000	0.51	1861	372	311	411	352	423	21.93
	.73	2682	438	308	434	417	----	30.46
	.80	3060	472	308	440	482	466	36.23
	.83	3541	491	312	459	590	502	44.27
	.99	4112	588	314	474	790	547	51.68
50,000	0.38	1379	273	247	410	263	410	17.39
	.48	1786	289	247	410	275	428	17.60
	.59	2101	312	247	428	291	450	24.40
	.62	2107	311	240	422	289	431	24.04
	.71	2582	341	245	434	326	462	29.21
	.81	2920	372	242	444	366	467	33.61
	.78	3091	362	243	445	386	439	35.91

[redacted]

TABLE III. - COMPRESSOR-OUTLET BLEED PERFORMANCE DATA

Altitude, ft	Free-stream Mach number, M_0	Engine speed, N, rpm	Compressor-inlet total pressure, P_2, lb/sq ft abs	Altitude static pressure, p_0, lb/sq ft abs	Compressor-inlet total temperature, T_2, °R	Compressor outlet: Total pressure, P_3, lb/sq ft abs	Compressor outlet: Total temperature, T_3, °R	Turbine inlet: Total pressure, P_4, lb/sq ft abs	Turbine inlet: Total temperature, T_4, °R	Turbine outlet: Total pressure, P_5, lb/sq ft abs	Turbine outlet: Total temperature, T_5, °R	Exhaust-nozzle inlet: Total pressure, P_8, lb/sq ft abs	Exhaust-nozzle inlet: Total temperature, T_8, °R	Manufacturer's temperature average, T_8, °R	Engine-inlet air flow, $W_{a,2}$, lb/sec	Compressor-outlet bleed-air flow, $W_{B,3}$, lb/sec	Engine fuel flow, W_f, lb/hr	Measured jet thrust, F_j, lb
37,500	0.8	6015	640	409	425	3119	716	2943	1211	960	940	920	929	958	38.89	0	930	1726
		6002	644	418	421	3126	706	2953	1196	966	926	924	914	942	39.22	0	930	1711
		6000	645	411	419	3105	700	2909	1223	958	949	915	934	983	39.48	1.31	954	1685
		6008	645	416	418	3117	705	2944	1240	956	970	913	955	977	39.82	1.59	991	1708
		6005	645	423	416	3066	702	2897	1262	942	980	901	966	987	39.71	2.20	999	1659
		6003	645	417	418	3049	701	2878	1261	937	982	896	965	990	39.58	2.38	999	1647
		6005	643	416	420	3029	704	2859	1264	938	982	895	966	996	39.34	2.36	989	1605
		6008	641	412	421	3008	708	2835	1265	925	983	882	967	995	39.08	2.42	985	1585
		6002	646	419	420	3055	697	2849	1248	932	971	895	956	985	39.53	2.47	989	1626
		6568	645	381	422	4191	778	3961	1505	1290	1162	1244	1163	1187	46.54	0	1641	2725
		6565	642	421	421	4175	780	3845	1504	1301	1165	1250	1164	1185	46.83	0	1635	2671
		6568	644	427	420	4087	775	3866	1544	1270	1201	1218	1198	1215	46.75	1.58	1684	2589
		6572	643	420	419	4068	774	3848	1558	1262	1212	1210	1206	1228	46.81	2.09	1696	2577
		6565	644	417	420	4064	773	3845	1568	1266	1221	1203	1215	1235	46.84	2.48	1721	2560
		6566	645	414	420	4035	768	3811	1571	1248	1224	1200	1220	1240	46.57	2.83	1721	2611
		6567	646	414	419	4044	771	3827	1578	1266	1230	1202	1226	1240	46.95	2.88	1729	2526
		6568	644	414	419	4023	769	3805	1583	1261	1236	1196	1231	1245	46.90	3.15	1734	2615
		6558	644	397	421	4045	786	3786	1575	1244	1231	1197	1226	1246	46.50	3.14	1727	2545
		7064	642	409	416	4907	823	4659	1690	1526	1315	1667	1314	1530	51.32	0	2205	3324
		7059	641	422	415	4824	818	4560	1725	1497	1340	1438	1345	1560	51.25	1.42	2251	3219
		7058	641	417	417	4767	818	4509	1749	1481	1556	1429	1560	1575	51.09	2.04	2251	3178
		7058	641	414	419	4891	819	4435	1760	1467	1570	1408	1577	1590	50.78	2.66	2264	3108
		7059	638	418	421	4649	819	4385	1778	1452	1583	1395	1568	1602	50.14	3.20	2271	3092
		7062	641	419	420	4658	818	4408	1795	1457	1389	1396	1594	1606	50.77	3.54	2297	3060
		7554	645	419	424	5384	887	5079	1915	1665	1496	1603	1496	1518	52.45	0	2745	3780
		7563	644	416	422	5299	878	4998	1963	1668	1535	1598	1526	1545	52.80	1.71	2821	3673
		7559	640	415	421	5193	873	4898	1970	1617	1554	1554	1543	1558	52.40	2.54	2829	3585
		7565	641	407	420	5160	871	4868	1988	1622	1565	1559	1557	1572	52.64	3.02	2844	3543
		7567	641	411	418	5141	868	4851	1992	1611	1566	1547	1561	1570	52.68	3.44	2856	3502
		7568	640	421	419	5112	865	4827	1991	1604	1572	1639	1564	1575	52.58	3.76	2844	3468
		7799	648	417	430	5630	919	5200	2078	1722	1625	1657	1630	1688	53.18	1.83	3126	3887
		7785	650	416	427	5402	907	5083	2098	1688	1680	1621	1658	1680	53.13	2.38	3141	3750
		7803	647	428	429	5458	914	5113	2104	1701	1689	1638	1657	1680	52.84	2.86	3165	3723

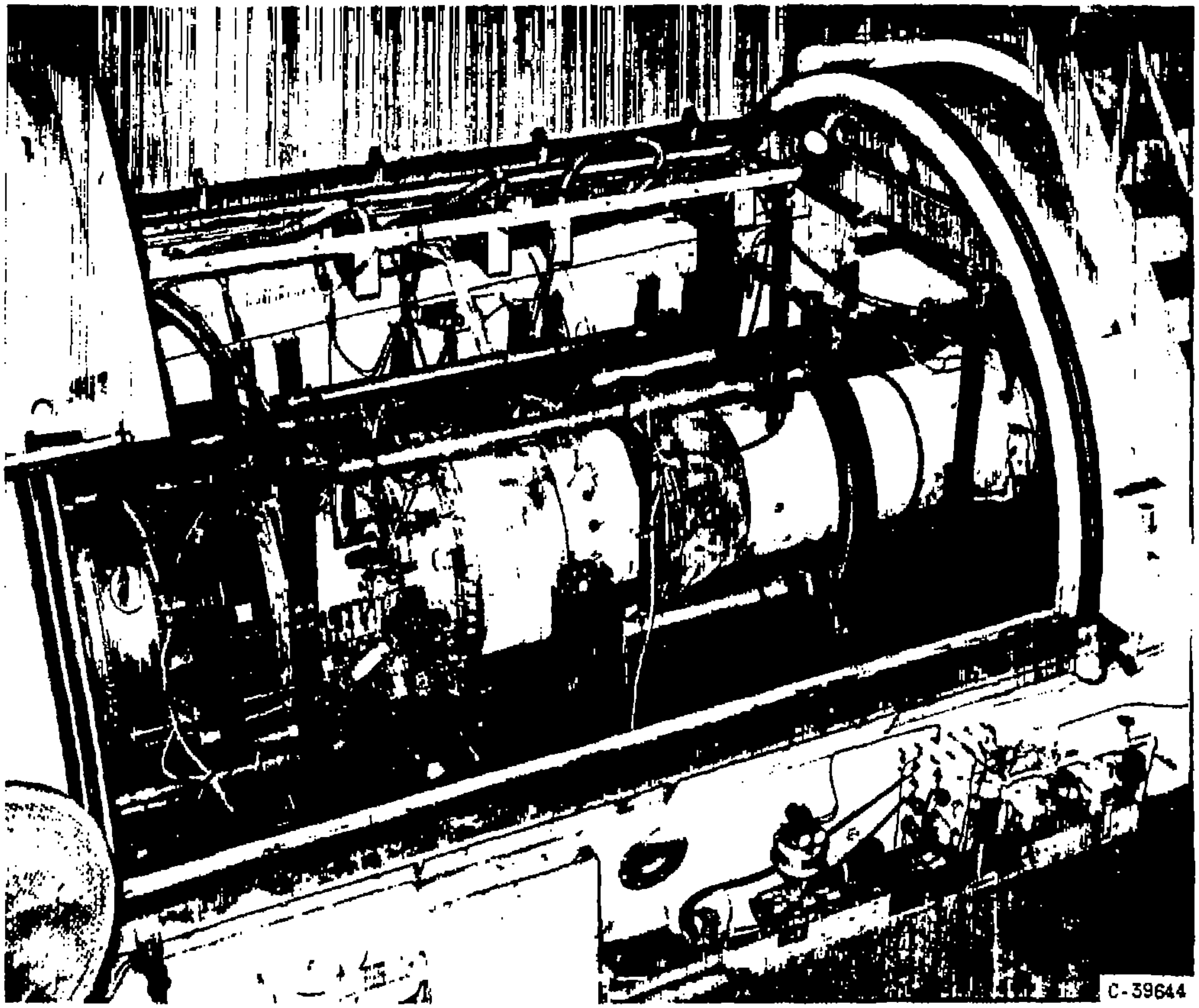
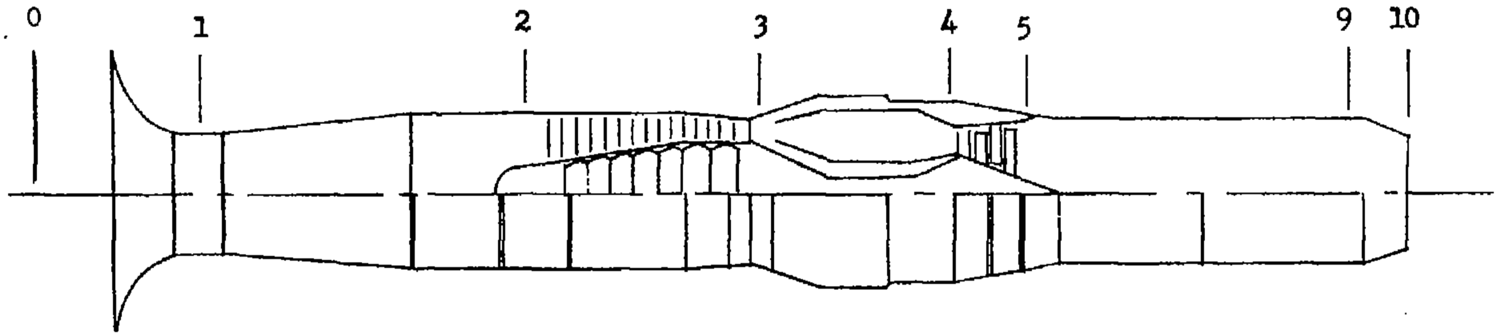


Figure 1. - RA-14 Avon turbojet engine installed in altitude test chamber.



Station	Number of probes		
	Static pressure	Total pressure	Total temperature
1	4	8 + 6 boundary layer	
2		18 + 12 boundary layer	9
3	1	20	24
4		12	28
5		24	20
9	3	16	16 + 4 skin
10	4		

Figure 2. - Schematic sketch of RA-14 Avon turbojet engine showing instrumentation stations.

[redacted]

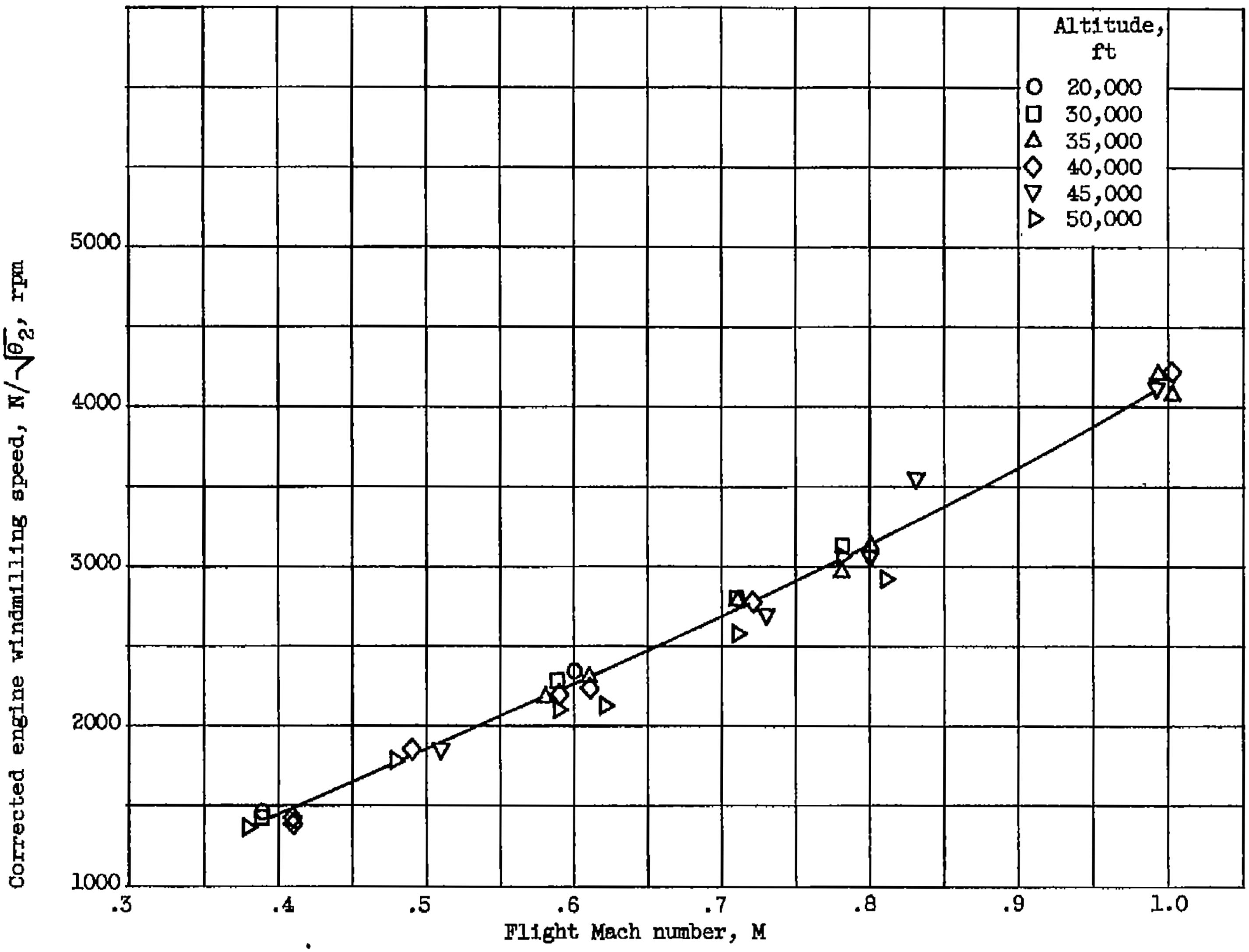


Figure 3. - Corrected engine windmilling speeds for range of altitudes and flight Mach numbers.

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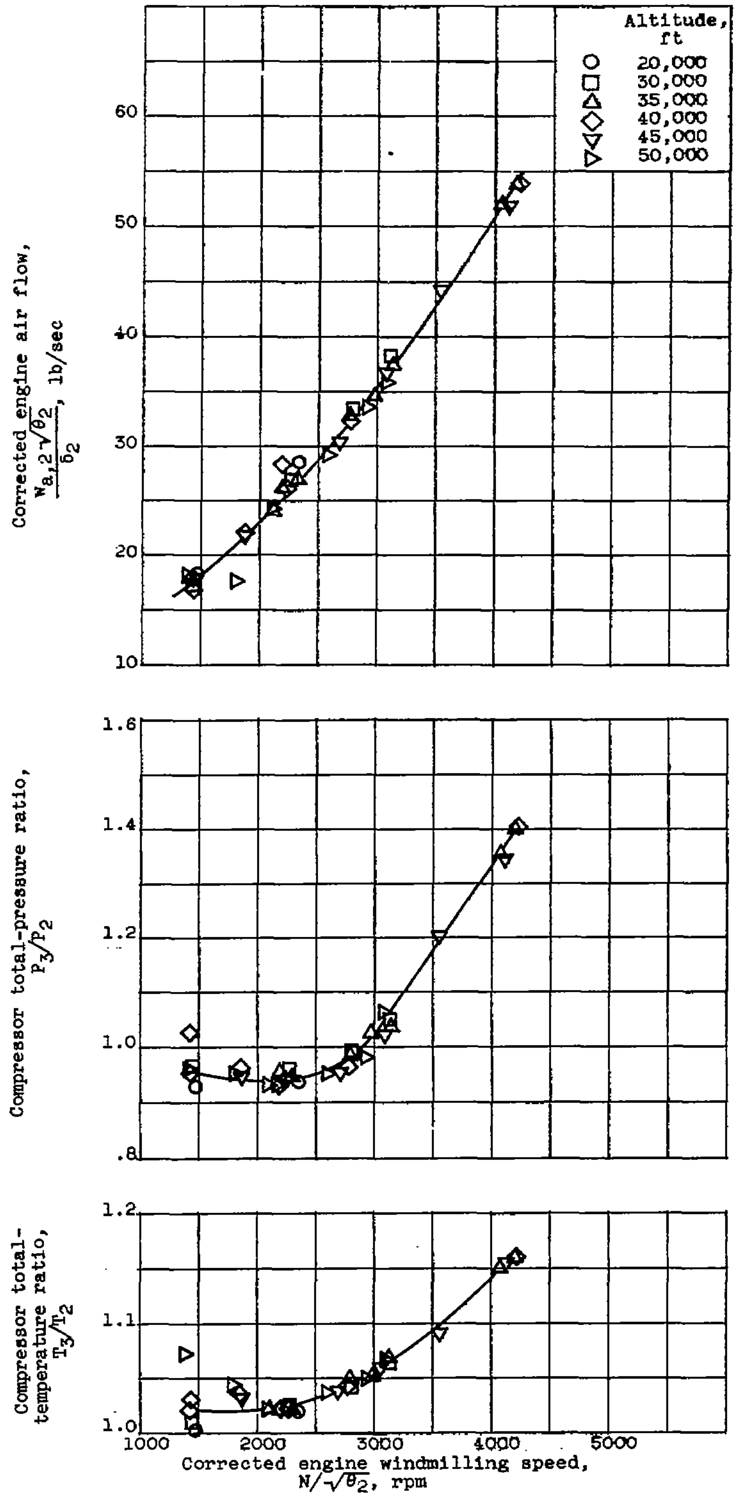

Figure 4. - Windmilling compressor performance.

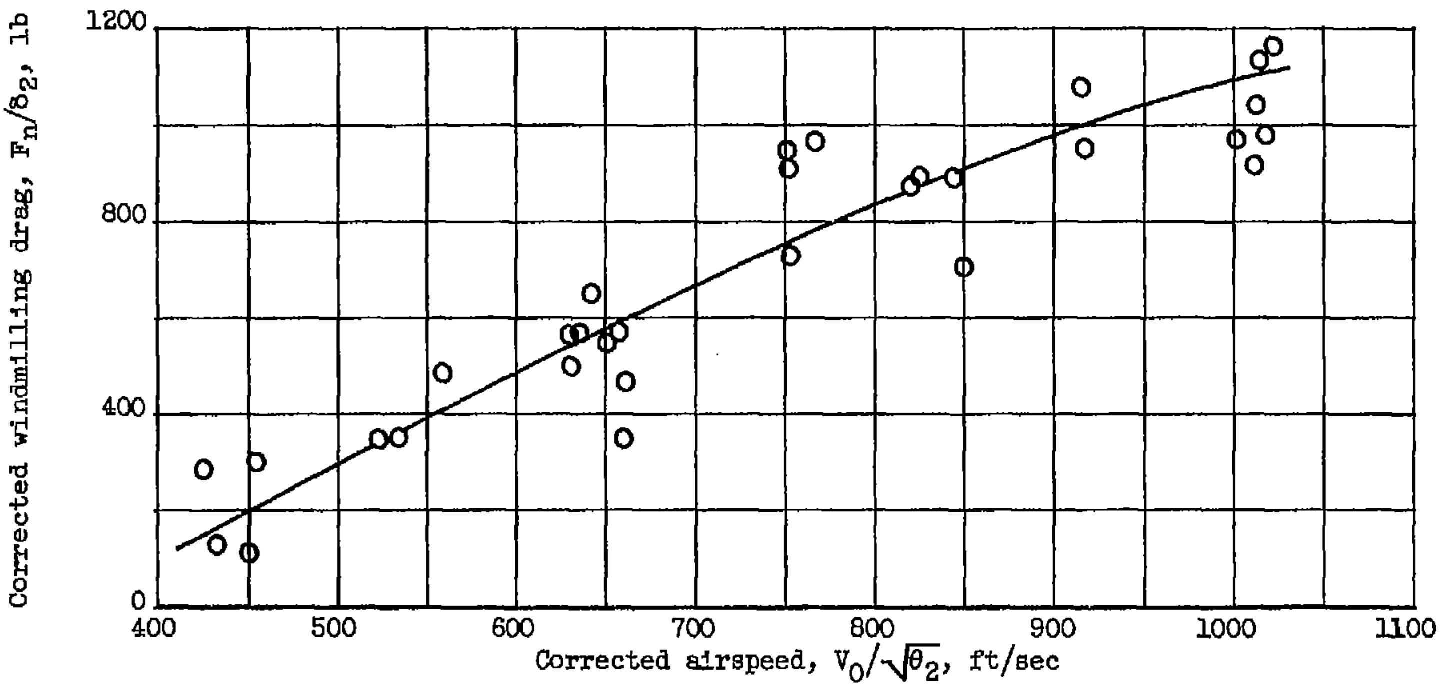

Figure 5. - Engine windmilling drag characteristics.

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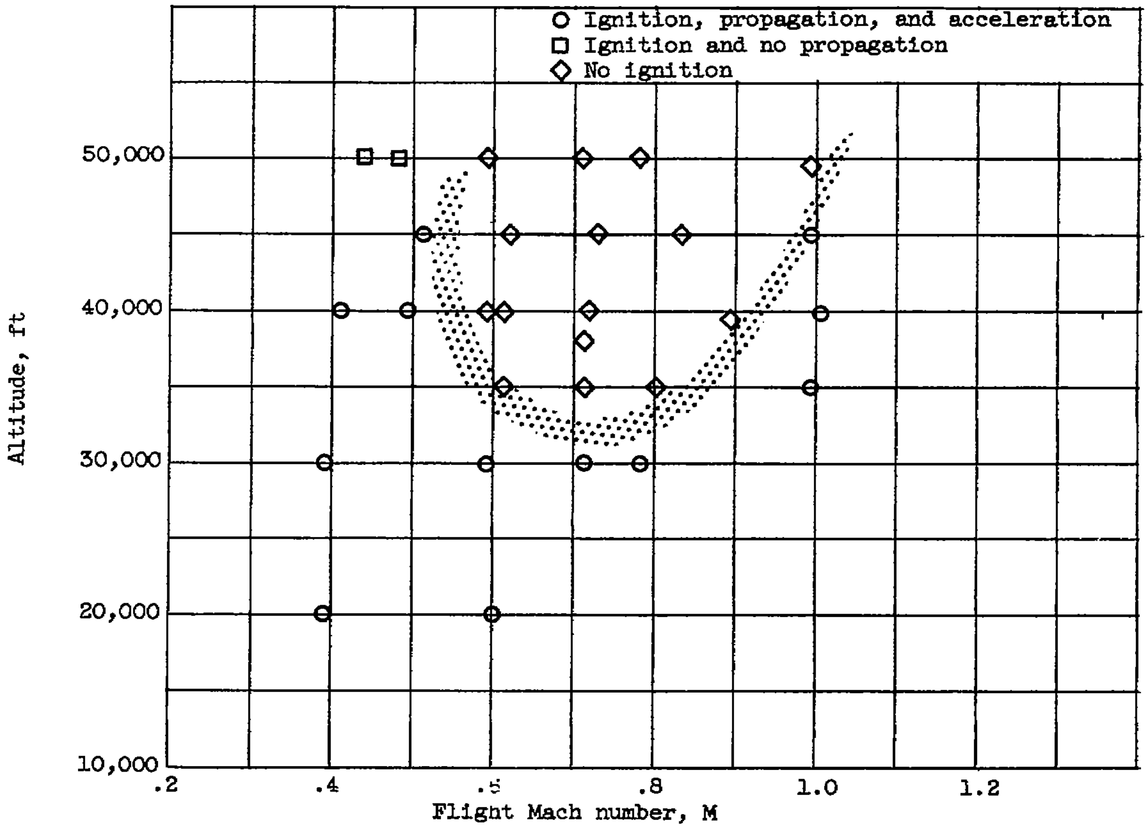

Figure 6. - Altitude starting characteristics of RA-14 Avon turbojet engine.

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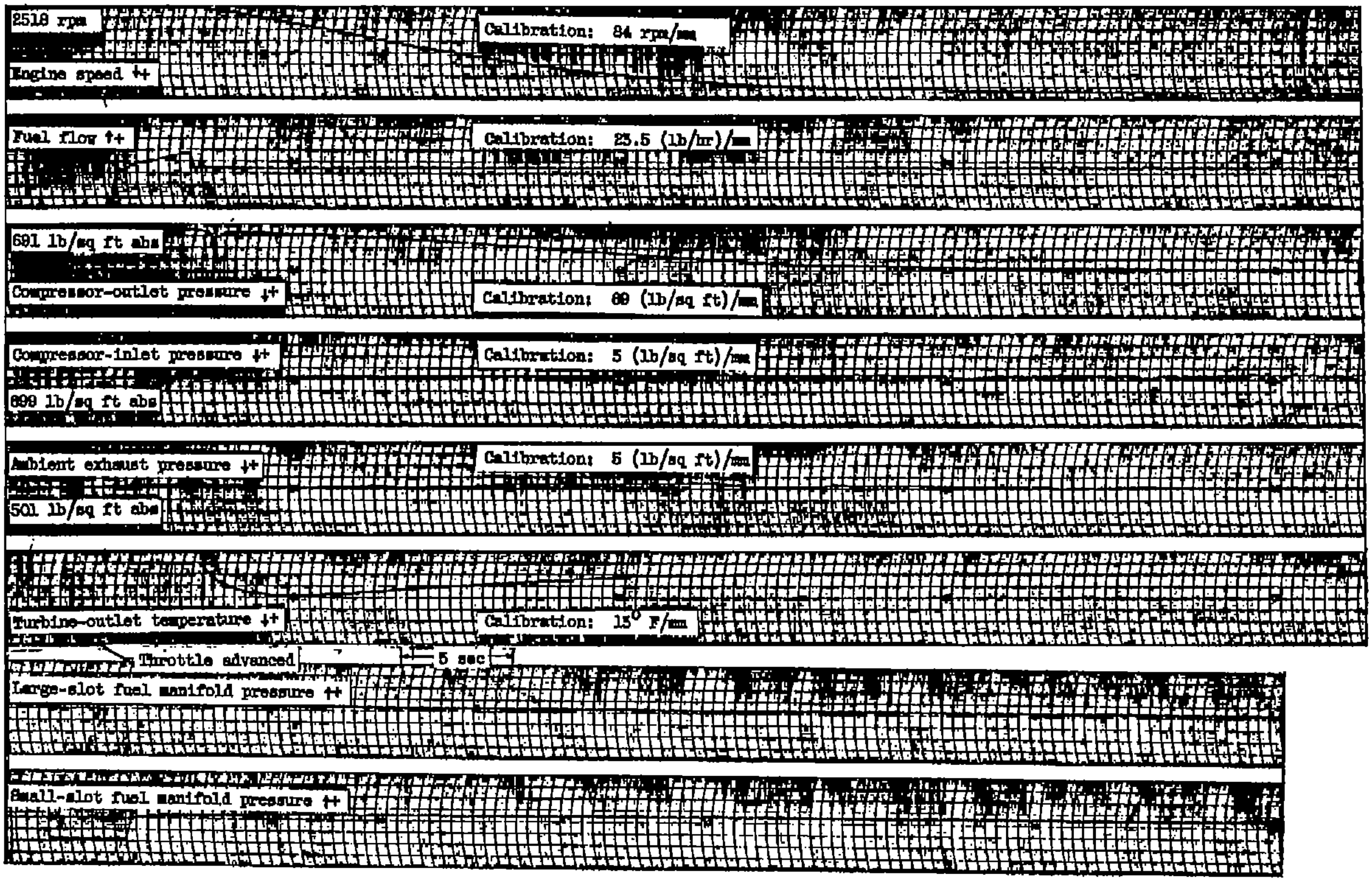

Figure 7. - Altitude ignition and acceleration to idle speed. Altitude, 35,000 feet; flight Mach number, 0.7.

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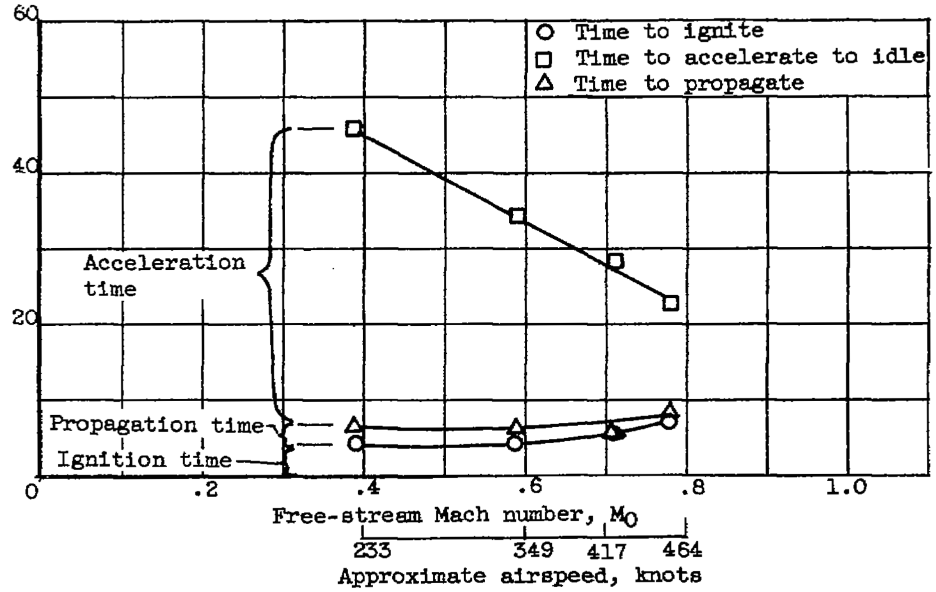
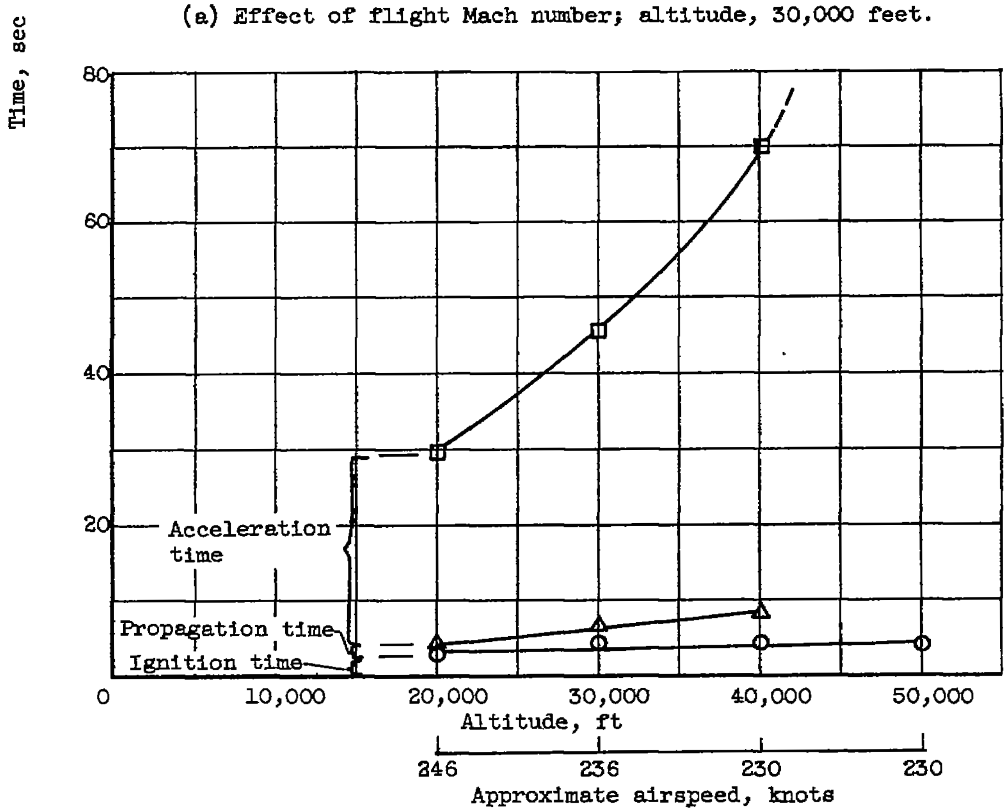


(a) Effect of flight Mach number; altitude, 30,000 feet.



(b) Effect of altitude; flight Mach number, 0.4

Figure 8. - Ignition, propagation, and acceleration characteristics.

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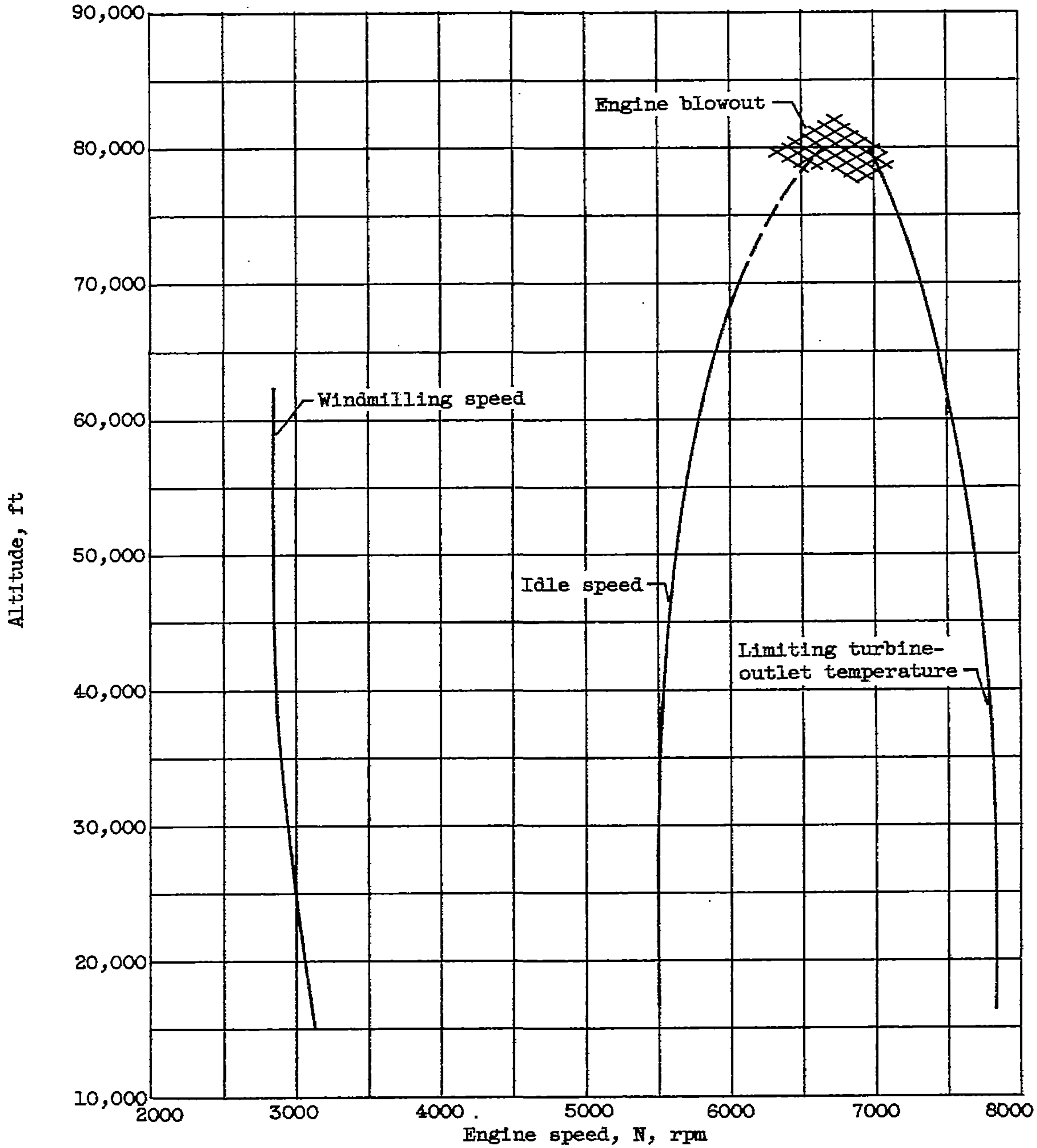

Figure 9. - Operational limits of RA-14 Avon turbojet engine. Flight Mach number, 0.8.

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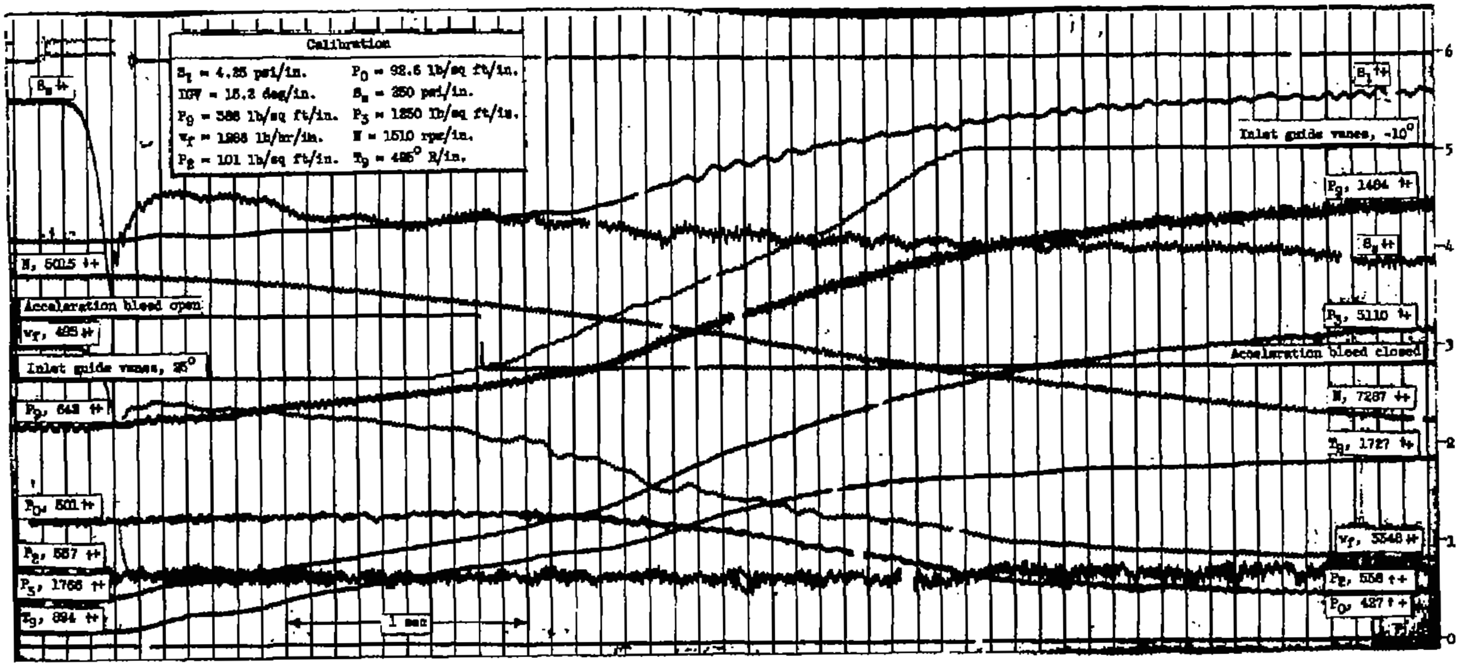

Figure 10. - Acceleration attempt from idle to rated speed (fig. 4(i), ref. 2). Altitude, 35,000 feet; flight Mach number, 0.4; no surge occurred.

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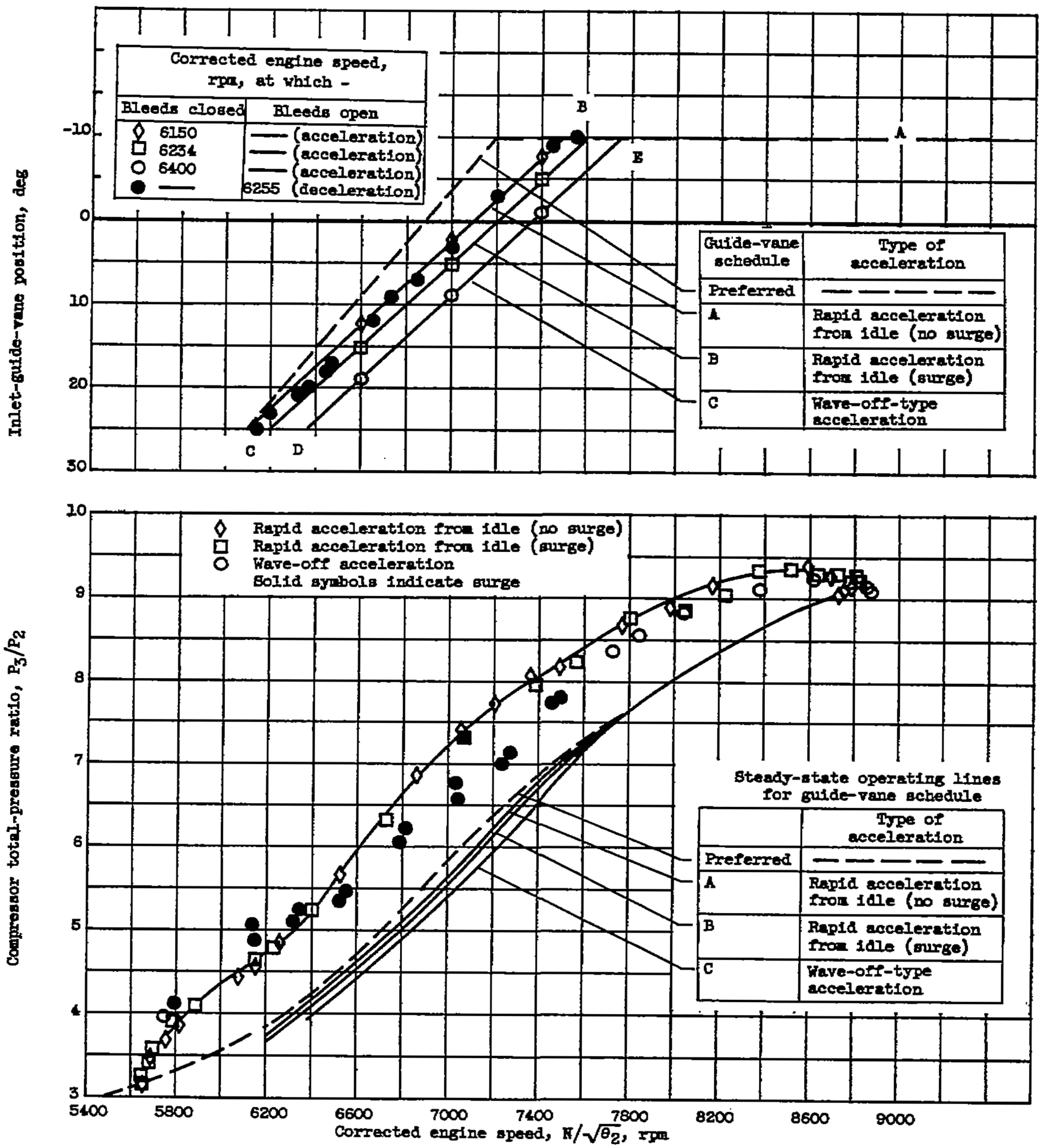

Figure 11. - Acceleration path comparisons at altitude of 35,000 feet and flight Mach number of 0.4 (figs. 9(a) and (b), ref. 2).

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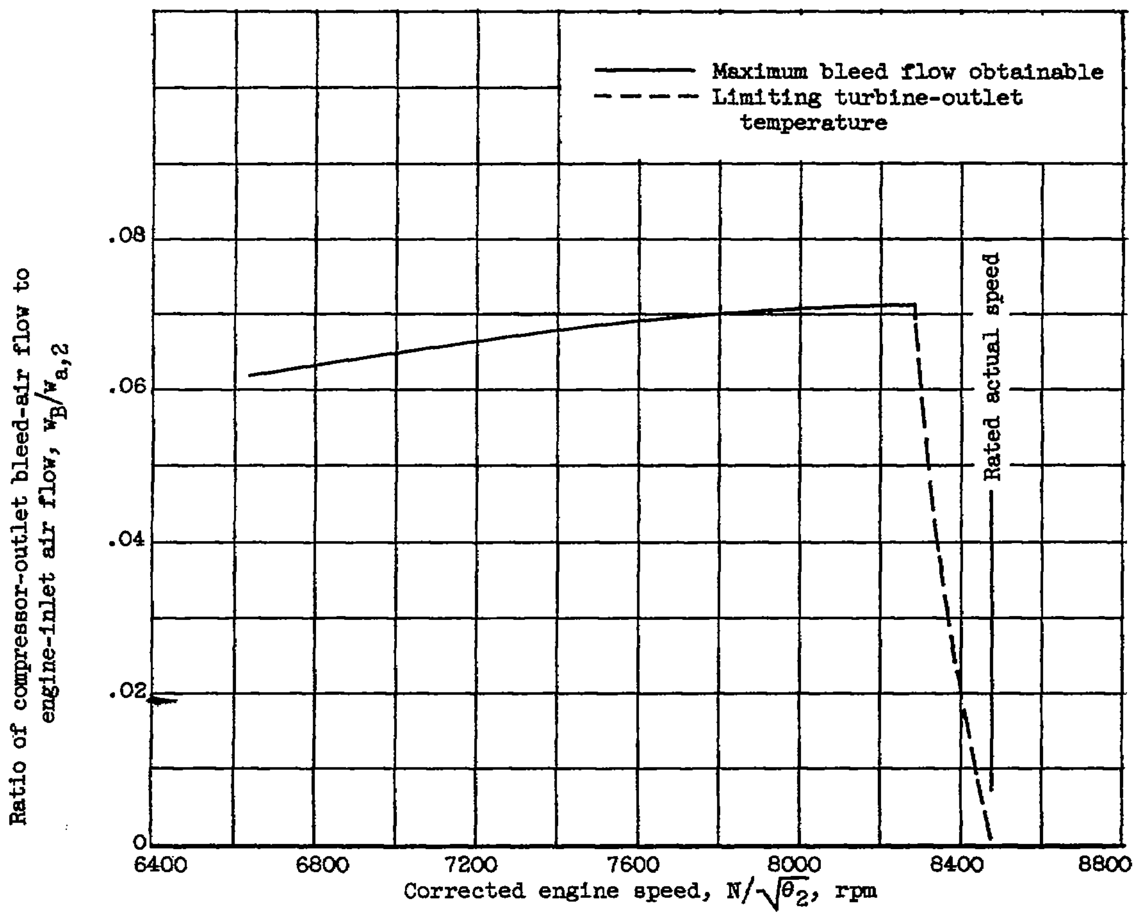

Figure 12. - Maximum obtainable compressor-outlet bleed flow. Altitude, 37,500 feet; flight Mach number, 0.8.

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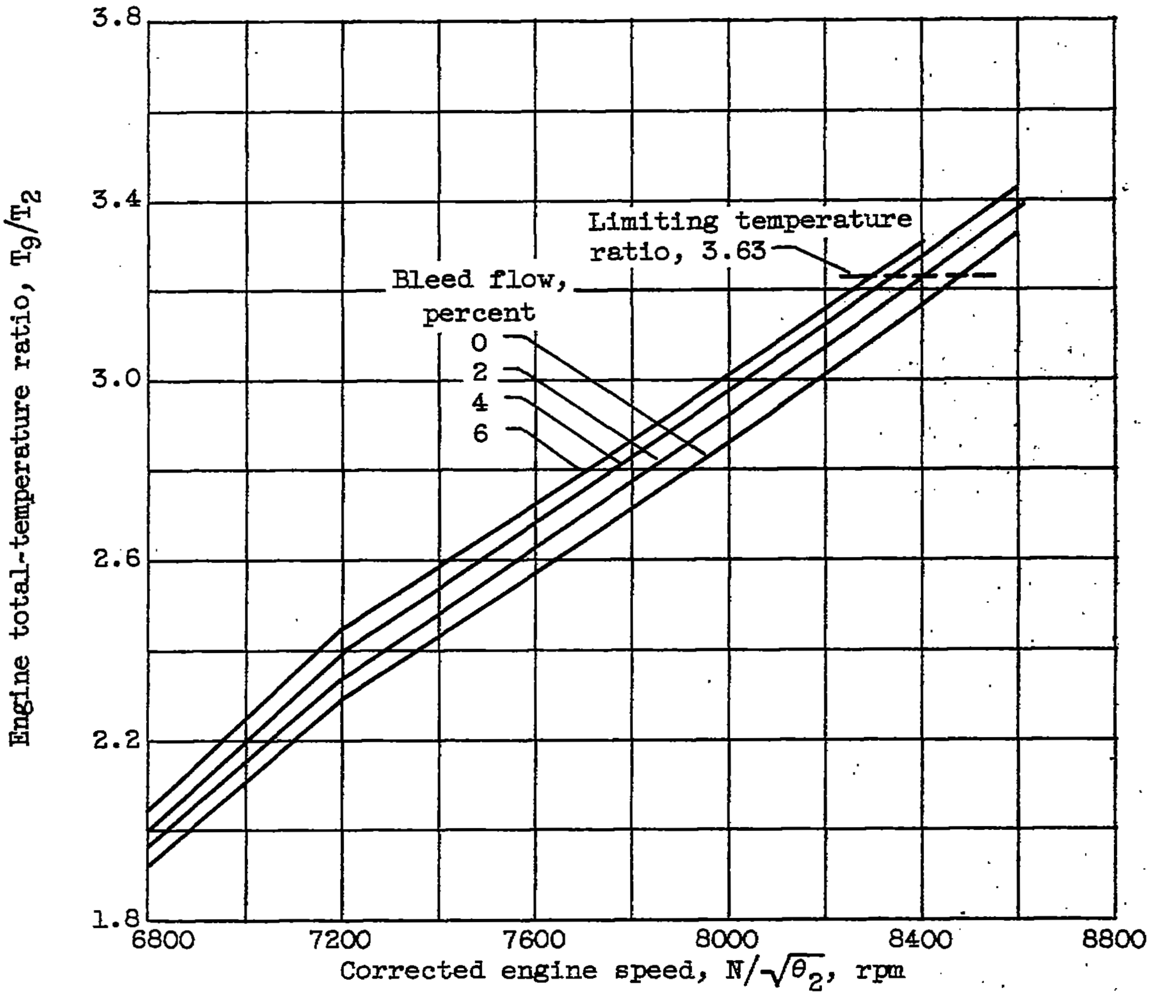

(a) Engine total-temperature ratio.

Figure 13. - Over-all engine performance characteristics for various compressor-outlet bleed flows. Altitude, 37,000 feet; flight Mach number, 0.8.

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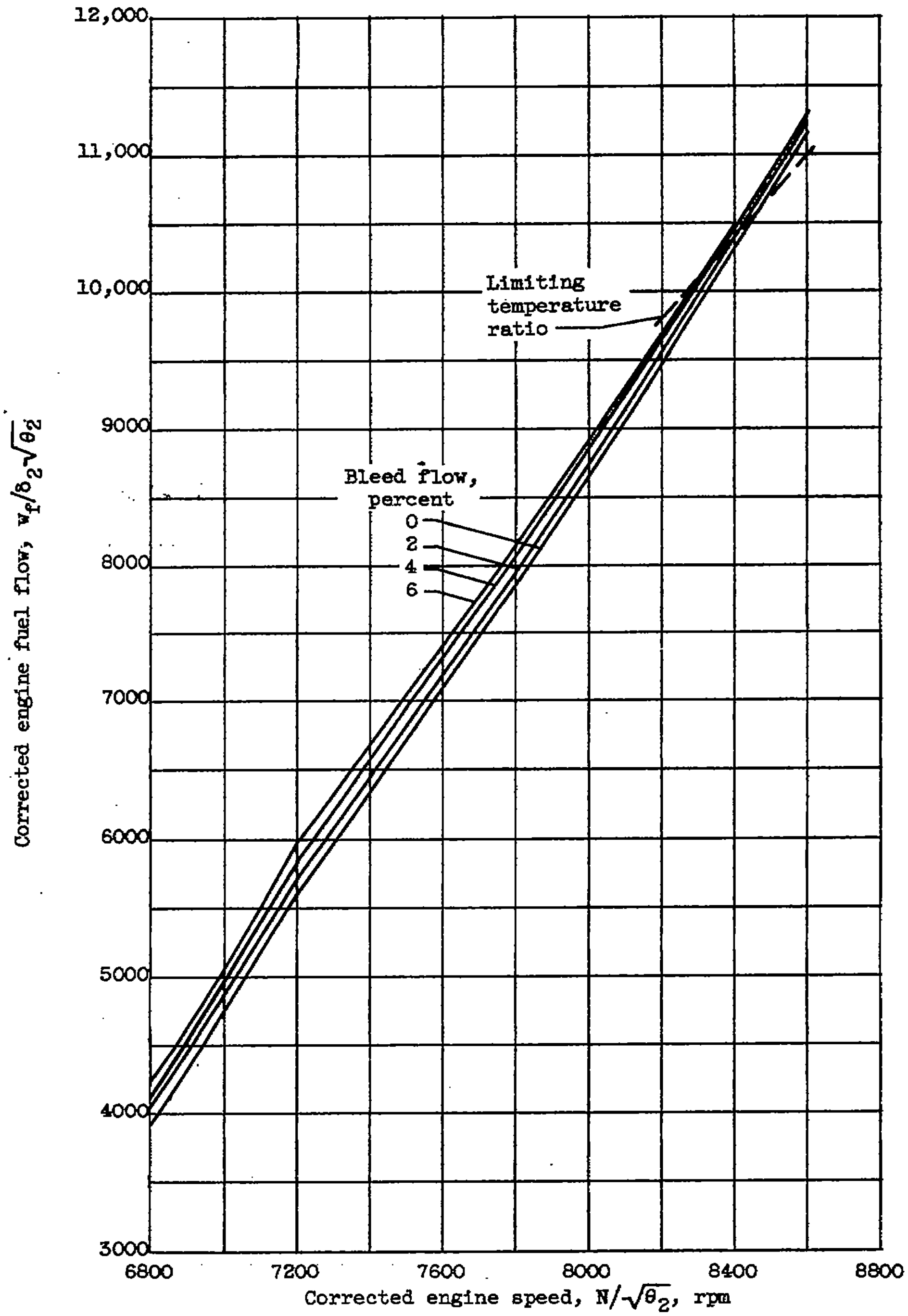

(b) Engine corrected fuel flow.

Figure 13. - Continued. Over-all engine performance characteristics for various compressor-outlet bleed flows. Altitude, 37,000 feet; flight Mach number, 0.8.

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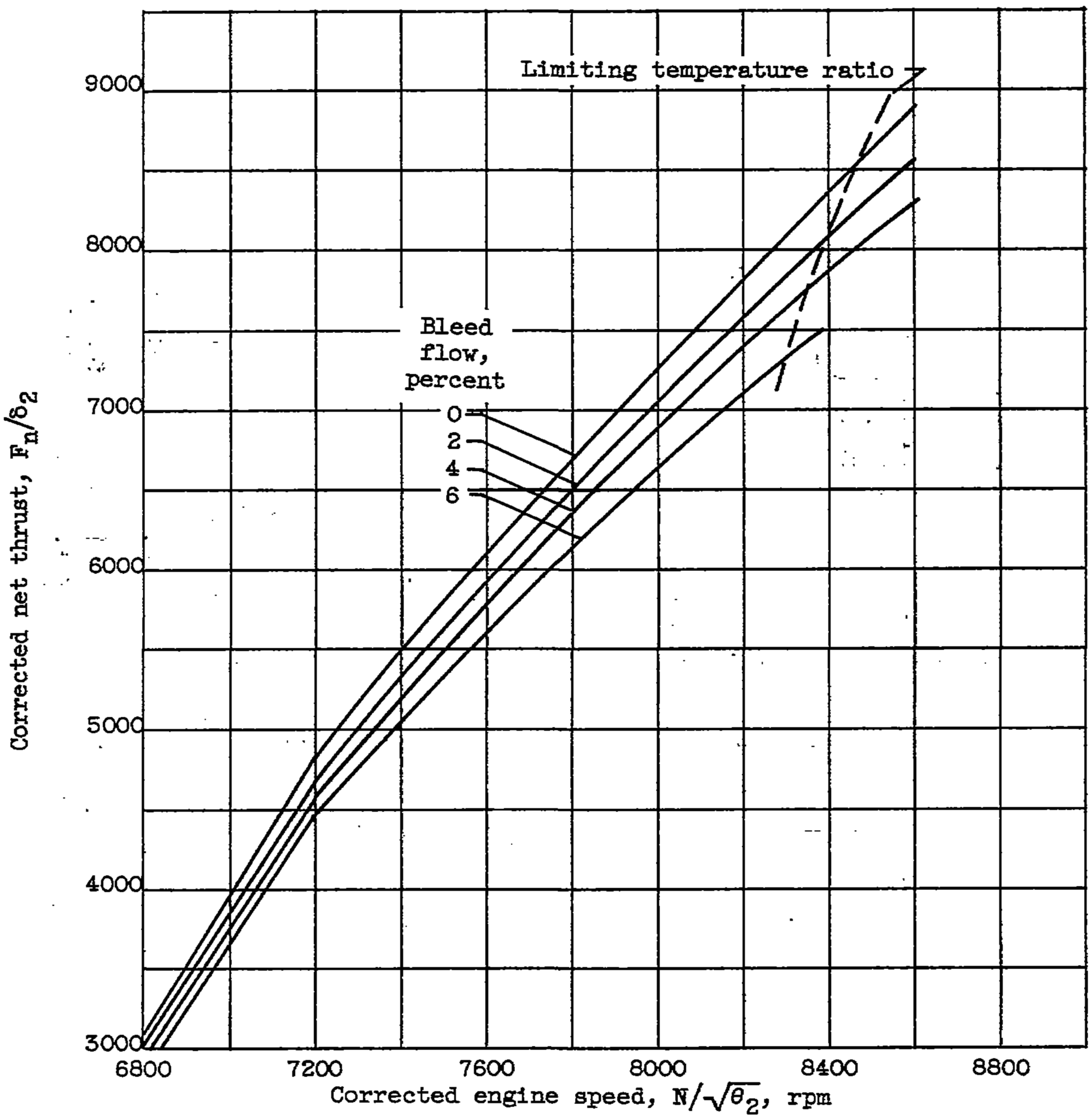
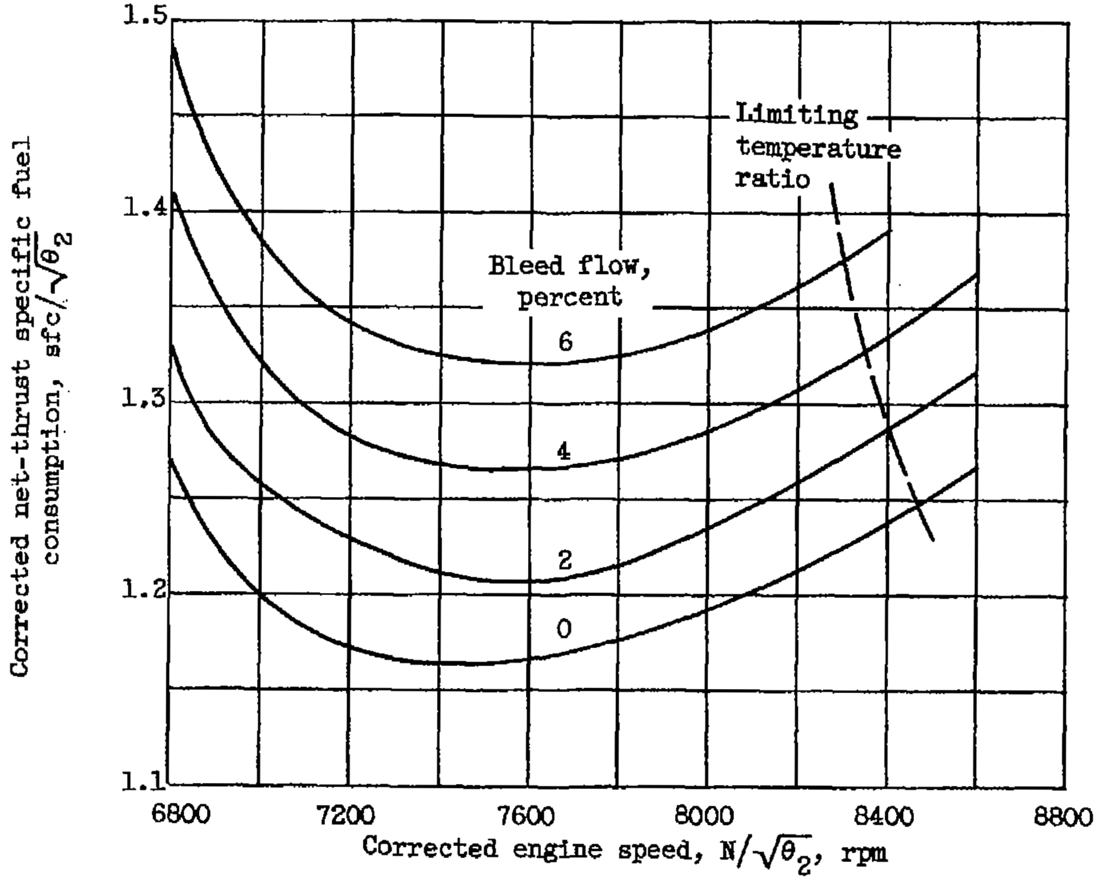


(c) Engine corrected net thrust.

Figure 13. - Continued. Over-all engine performance characteristics for various compressor-outlet bleed flows. Altitude, 37,000 feet; flight Mach number, 0.8.



(d) Engine corrected net-thrust specific fuel consumption.

Figure 13. - Concluded. Over-all engine performance characteristics for various compressor-outlet bleed flows. Altitude, 37,000 feet; flight Mach number, 0.8.

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OPERATIONAL CHARACTERISTICS OF RA-14 AVON TURBOJET ENGINE

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